AD-A126 909 FEASIBILITY OF ESTABLISHING A CORRELATION BETWEEN BOUNDARY LAYER GROWTH A..(U) SHANNON ENGINEERING SEATTLE WA O LETH ET AL. 30 SEP 82 JS1109 1/1

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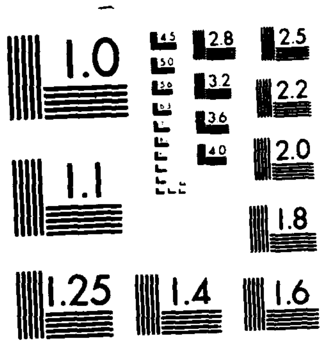

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Report FQ8671-00878

FEASIBILITY OF ESTABLISHING A CORRELATION BETWEEN BOUNDARY LAYER GROWTH AND UPSTREAM SHOCK CONDITIONS

AD A126909

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30 September 1982

Final Report for Period 16 April 1982 - 30 September 1982

Approved for public release; distribution unlimited

Prepared for:
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REPORT DOCUMENTATION PAGE

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1. REPORT NUMBER AFOSR-TR- 83-0196	2. GOVT ACCESSION NO. AD-A126	3. RECIPIENT'S CATALOG NUMBER 909
4. TITLE (and Subtitle) FEASIBILITY OF ESTABLISHING CORRELATION BETWEEN BOUNDARY LAYER GROWTH AND UPSTREAM SHOCK CONDITIONS		5. TYPE OF REPORT & PERIOD COVERED FINAL 16 April 82 - 30 Sep 82
		6. PERFORMING ORG. REPORT NUMBER JS1109
7. AUTHOR(s) OLE LETH JACK SHANNON		8. CONTRACT OR GRANT NUMBER(s) F49620-82-C-0059
9. PERFORMING ORGANIZATION NAME AND ADDRESS SHANNON ENGINEERING 215 - TERMINAL BLDG, BOEING FIELD SEATTLE, WA 98108		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS 61102F 2307/A1
11. CONTROLLING OFFICE NAME AND ADDRESS AIR FORCE OFFICE OF SCIENTIFIC RESEARCH/NA BOLLING AFB DC 20332		12. REPORT DATE 30 September 1982
		13. NUMBER OF PAGES 27
14. MONITORING AGENCY NAME & ADDRESS(if different from Controlling Office) [redacted]		15. SECURITY CLASS. (of this report) UNCLASSIFIED
		15a. DECLASSIFICATION DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

VISCOUS/INVISCID INTERACTION
DRAG
BOUNDARY LAYER
BOUNDARY LAYER SEPARATION
TRANSONIC AIRFOIL
EXPERIMENTAL
HIGH REYNOLDS' NUMBER

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

An extensive search in the industry and within the academic community to identify data suitable for drag correlations studies was conducted. No single data set or combination thereof were found to be adequate for analytic treatment. It was concluded that the technique of simulating higher Reynolds' number effects by boundary layer transition strips is unsuitable for drag correlation studies. An economical test sequence using existing airfoil types in high Reynolds' number tunnels was recommended.

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SUMMARY

The emphasis on more efficient and higher performance aircraft has precipitated radical changes in airfoil shapes and planforms. These changes have the potential to produce superior aircraft performance at or near the design Mach and lift coefficient and the improvement has generally been predicted using current engineering procedures. However, one characteristic, drag creep or premature drag rise with Mach Number, has been difficult or impossible to predict. Often new aircraft performance is marred by poor drag characteristics at, or near, the design point.

The purpose of this work was to conduct a literature search and locate test data suitable for analysis to investigate boundary layer and shock interactions contributing to premature drag rise.

It has been determined that an insufficient test data base exists to satisfy that purpose.

The data search and evaluation has included contacting the industry, the academic community and government institutions, and reviewing all accessible and qualifying material. To narrow the scope of the effort to manageable proportions, two-dimensional data was emphasized although three-dimensional data was collected in the process.

It was not surprising that most material from the private sector was classified as proprietary data. Significant data was found in the work by McDonnell Douglas Research Laboratories and summarized in NASA TM-81336, which is unpublished. This work was near the comprehensive nature required for the analysis attempted in this report. However, even this package did not have the matrix of data necessary for this study.

It is significant that the above material allowed one important conclusion: that Reynolds' number effects on drag cannot be studied using an artificial boundary layer transition. This technique is commonly used in aerodynamic testing.

Based on the assessment that limited data is available for analysis and that specific test at high Reynolds' numbers are needed, it is recommended that a test program combined with a continuation of the work summarized in NASA paper TM-81336 be undertaken. This would be a cost effective means of continuing with this research.

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A

A

SECTION I

INTRODUCTION

1. Objective

The purpose of the work contained in this report was to attempt to bridge the gap between the empirical and the purely analytical approach by correlating drag rise to underlying physical parameters in the boundary layer. The means to meet this objective was to identify data suitable for analysis to investigate boundary layer and shock interactions contributing to premature drag rise.

2. Historical Review

Early predictions of vehicle aerodynamic drag were for the most part empirical. Historically, work by Horner (Reference 8), which is largely a compilation of discreet experimental drag results of two-dimensional or axisymetric objects, is and has been universally accepted as a basis for drag predictions. Complex three-dimensional configurations have been tested by geometrically scaled objects or models in flow fields similar to the full scale conditions.

The advent of near sonic flight, which occurred in the nineteen forties, taxed traditional drag prediction techniques beyond limits of reliability. In this flight regime local sonic flow and the boundary layer interact to alter global flow fields. The resulting aerodynamic characteristics differ significantly from levels predicted by simple boundary layer growth and first order shock location studies.

There are several examples (Reference 9), of airplanes which have experienced premature Mach drag onset. Such errors in drag predictions have, from their earliest occurrence, typically been treated as an individual and unique problem. Solutions have included the use of vortex generators, local aerodynamic tailoring, and leading edge or trailing edge camber modifications to the airfoil.

The frequency and severity of prediction errors have increased in recent years as a result of the trend toward thicker wings operating at high design lift coefficients and Mach number.

The error can be significant. Data from Reference 10 suggest a 10 percent premature transonic drag rise well below the Mach and lift coefficient design point. Conversely, an impressive drag reduction of over 10 percent, such as demonstrated in Reference 11, has been obtained by careful attention to the pressure signatures in areas of critical local sonic flow.

A notable miss in the prediction of trailing edge separation and hence airplane drag rise due to Mach number is recorded in Reference 12. Wing shock locations were noted in flight test that missed wind tunnel predictions by as much as 20 percent wing chord. Large discrepancies in lift (and hence drag) and pitching moment resulted. Extensive work by the authors led to the definition of a $B^{1/2}$ term which promised to establish boundaries for a wing to preclude the effects of trailing edge separation. However, Figure 1 shows the results of including two new airplanes, the CL600 and the Lear 24, in this data set. Trailing edge separation occurs earlier than predicted on the CL600 and does not separate as early and actually contradicts the predicted trend at one station on the Model 24 Learjet.

Common to all of these experiences is the full scale flight testing followed by aerodynamic tailoring. This after-the-fact understanding and solution process has become an accepted procedure in aircraft development and design.

The limitations in a purely empirical approach has prompted others to resort to semi-empirical techniques. Shevell has in Reference 6 succeeded in providing a tool for predicting the drag divergence Mach number and incremental drag coefficients due to compressibility on wings for modern air carriers. The technique appears to be accurate enough for most preliminary design work on aircraft of this type.

REF. 12: $B_{\frac{1}{2}} = \dfrac{P_{SAMB} / P_{SPEAK} \sqrt{M_{\infty}}}{\sqrt{1-(X_{SHOCK} / C_{WING})}}$

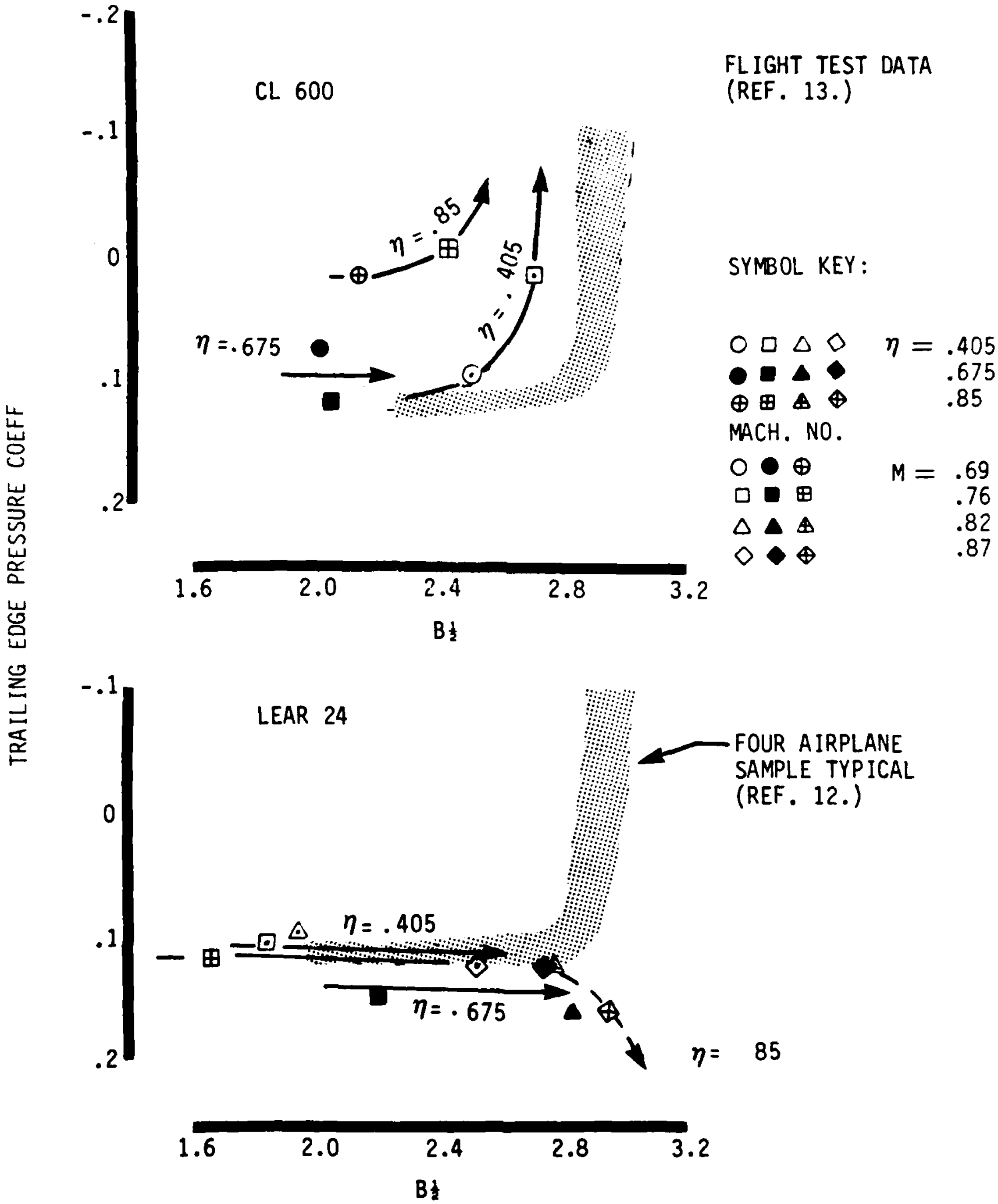


Figure 1. Trailing Edge Pressure Correlation

Figure 2 shows a very encouraging match of drag rise data when the subject method is applied to the sample data base, in this case two DC-8's, a DC-9 and a configuration identified as WB-1 (Wide Body 1).

Later data introduced in Figure 3 shows "Wide Body" data in which agreement is exhibited in only one case out of three. The larger disagreement is attributed to local sonic flow on the underneath side of the wing due to the large amount of wing twist.

The method is semi-empirical and retains a high level of success in predicting drag rise for configurations near the data base. However, it is questionable whether this work can be generally applied to other aircraft or profile configurations.

The complexity of the subject phenonmenon has stimulated the development of comprehensive analytical tools, such as that presented in Reference 4. Despite these capabilities, calculated transonic drag values are not reliable for design. The main contribution from such tools is the identification of high drag conditions via the calculation of separation points on the airfoil using the criterion developed by Stratford in Reference 5. This criterion is a first order expression relating the local velocity with the rate of change of the pressure in the boundary layer. Although separation is recognized as the condition responsible for premature drag rise, it is not as yet possible to accurately relate the predicted separation to a level of drag on the airfoil, or the onset of premature Mach drag.

The status of drag prediction technology is at first glance surprising considering the very sophisticated analytical tools available today and the extensive testing that is employed in an aircraft development programs. Also, work on two-dimensional models in the last few years has produced impressive results. Yet the full scale validation of these technologies demonstrates that drag creep for a transonic configuration cannot be predicted with satisfactory accuracy. Airfoil camber and

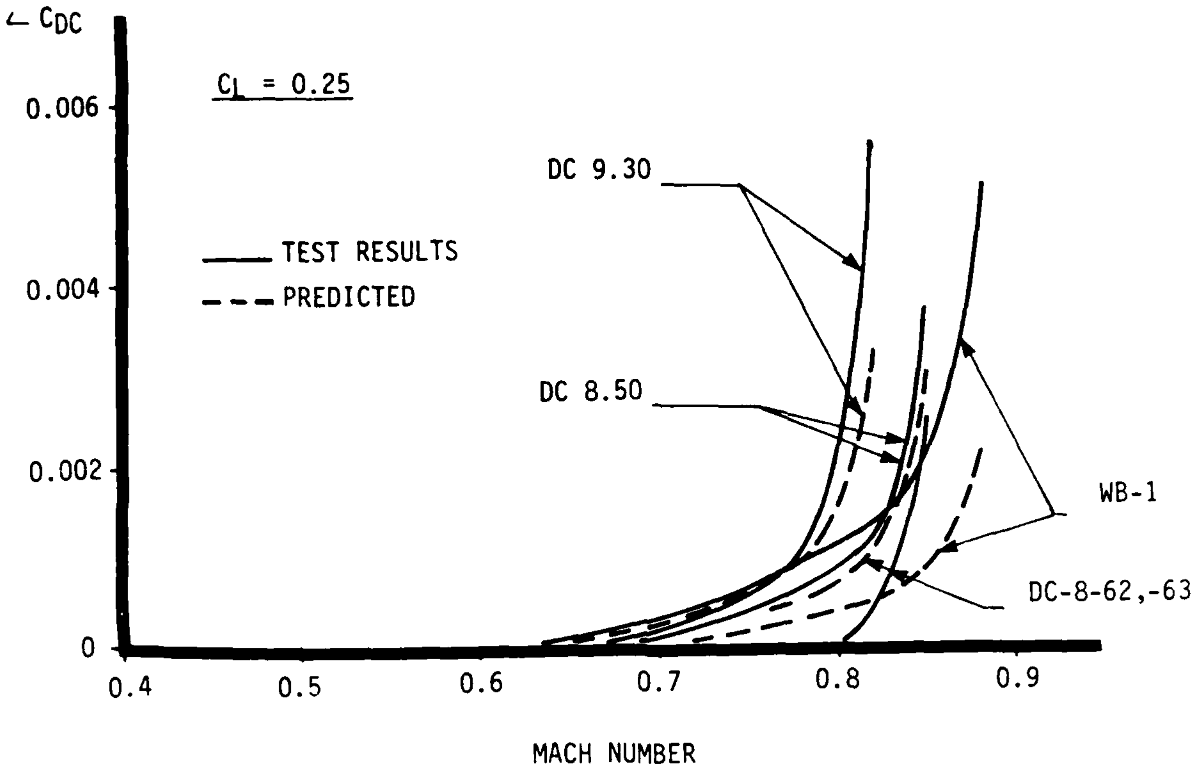


Figure 2. Test Results vs Predictions From Reference 6

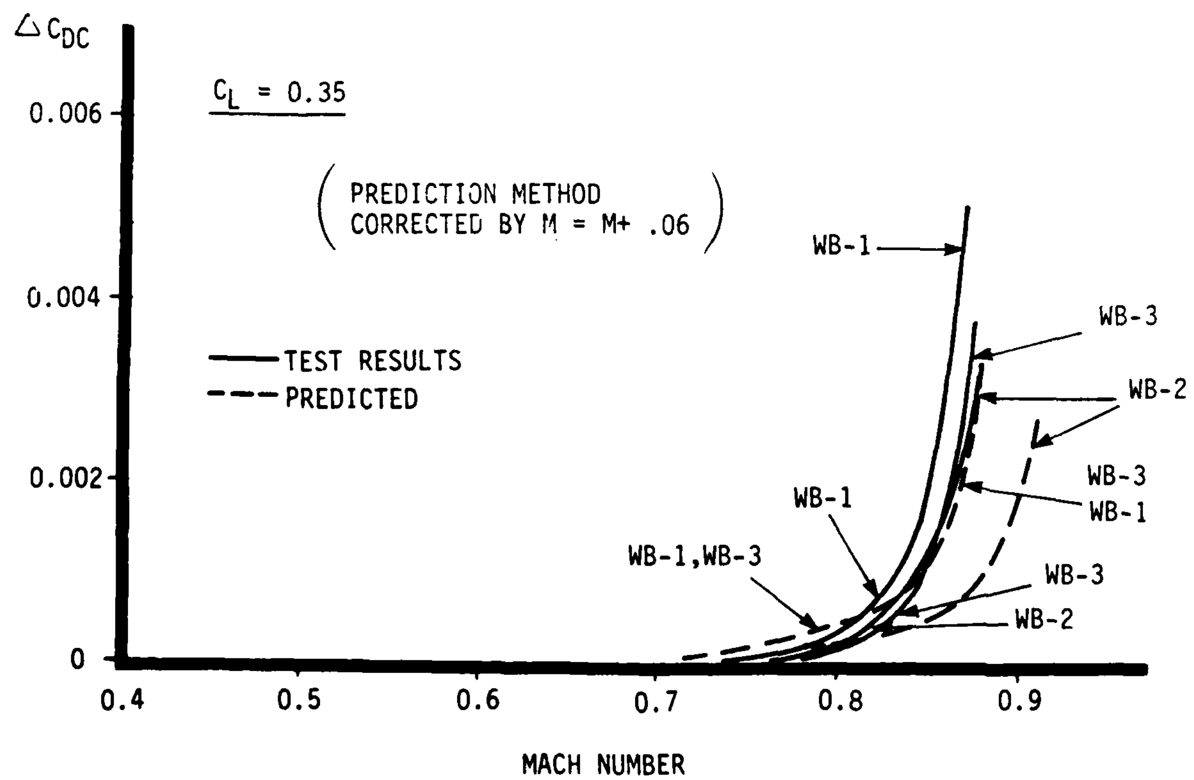


Figure 3. Test Results vs Predictions From Reference 6

thickness distributions well removed from a proven data base and especially three-dimensional effects make predictions difficult.

3. Aerodynamic Principles

The underlying phenonmenon responsible for drag creep and, the discrepancies in its prediction, is viscous/inviscid interactions, primarily the shock and boundary layer interactions on the upper aft surface of the airfoil. Reference 2 eloquently describes various forms of interactions that are distinct and have different dependence on Mach and Reynolds' number. Two flow categories or conditions are identified and allow for at least a qualitative description of specific cases. The traditional categorization is to label the flow an A Flow or a B Flow depending on whether the flow is characterized by Mach induced effects, or on classical trailing edge separation and consequently Reynolds' number effects. Thus, an A Flow is distinct by its tendency to develop separated flow behind the shock, and B Flow is typified by boundary layer separation at the trailing edge of the airfoil.

Schematics of these idealized flow patterns are included in Figure 4.

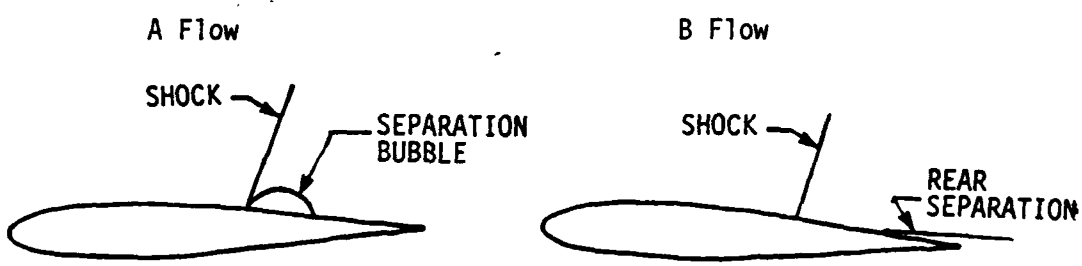


Figure 4. A Flow shown to the left, is characterized by shock induced separation behind the shock. B Flow, to the right, is characterized by separation starting at the trailing edge.

Mach and Reynolds' number effects are important parameters in the characterization of these flows. The A Flow is primarily sensitive to the local Mach number. Thus, from Reference 3 it is worthwhile to note that for A Flow the boundary layer will not separate for local Mach values $M_L < 1.3$ and separation is incipient for local Mach values above this level. Surprisingly, it has been found that this threshold is not considered for many airfoil designs, perhaps because some airfoils or their use dictate operation where A Flows may develop. As will be seen later, the design Mach number and local Mach are significant factors in defining airfoils with thick sections or high design lift coefficients. The B Flow on the other hand is primarily sensitive to Reynolds' number and of course lift on the airfoil which affects aft body pressure signatures.

The A and B Flows are idealized conditions. Modern airfoils normally have a high aft loading and are also operated at high Mach numbers. Depending on the type of flow, separation can either start or have a tendency to start at the shock and extend towards the trailing edge and thus cause total separation, A Flow, or separation may tend to start from the trailing edge and extend forward to the shock, B Flow. In either case, separate or combined flow patterns are involved resulting in off nominal airfoil performance and, most often, premature drag rise at operation near the design Mach and lift coefficient.

In addition to the local effects of Mach and Reynolds' number, it was found that the so called global or total flow field influences drag characteristics. A discussion will be presented in the data analysis section of the validity of using boundary layer transition strips where local sonic flow is present. As will be seen, it is important to test at full scale Reynolds' number in order to produce results valid for full scale characteristics.

Based on the above observations and the understanding of the boundary layer behavior and its importance in dictating the drag characteristics of a given airfoil, it was possible to outline a set of preliminary analytical goals and techniques to identify the data sample necessary for this study. The approach was to plot drag versus lift and Mach conditions, and to monitor the velocity profile on the upper trailing surface of the airfoil. Stratford's formula for separation would be applied and compared with actual measured velocity profiles. It was also the intent to monitor local Mach numbers in the free stream forward of the shock as the drag profiles developed.

For this purpose, the success of identifying suitable test data depended on the availability of data with boundary layer profile measurements in terms of velocity profiles, and simultaneous wake momentum measurements together with pressure signature data. It was decided that, due to the complexity of the phenonmenon, the study would be limited to two-dimensional conditions although three-dimensional data would be collected when available.

After an exhaustive search of data, which is chronicled in the next section, it was concluded that such data did not exist.

SECTION II

RESULTS

1. DATA SEARCH

The primary obligation under this contract was the identification and collection of test data suitable for a shock and boundary layer interaction study. In order to ensure that a maximum number of potential sources would be contacted, a program involving a literature search as well as written and direct contact of private industrial entities, government groups and universities was implemented. The academic community, was responsive; most noticeable Ohio State University Reserach Foundation and Stanford University, Department of Aeronautics and Astronautics. All provided data, documents and material pertinent to this study. Government groups were helpful and provided comments and advice on state-of-the-art tools in terms of analytical techniques, test facilities and procedures. In particular, the Langley group offered valuable information. Dryden Flight Research Facility under Ames Research Center contributed directly by providing full scale aircraft data. For obvious reasons, very limited response was received from the private sector. Competition in the industry unquestionably makes drag prediction technology a guarded trade secret. One private institution, however, McDonnell Douglas Research Laboratories, Saint Louis, turned out to be the most valuable contributor to the program as will be apparent from the subsequent chapters.

The library of test data compiled by Shannon Engineering over the years also served as a source of data in this study.

Although some leads pointed toward European studies on the subject of viscous/inviscid interactions, time constraints left the pursuit of these sources to future studies or a Phase II effort to this program.

A summary of the most pertinent material acquired or identified is presented in Table 1. The suitability of the material for the subject study is indicated. In order to ensure that the latest pertinent material would be identified, all of the entries at the AIAA/ASME 3rd Joint Thermophysics, Fluids, Plasma & Heat Transfer Conference in Saint Louis during the month of June this year, were examined. Reference 7 was found to include a comprehensive summary of the high Reynolds' number testing conducted at the Langley facility.

In order to gain a perspective on Table 1; five industrial firms, seven government institutions, and ten principals at Universities including research organizations were contacted in this study. Sixty-two documents or unpublished works were reviewed over a five month period to produce the summary of data in Table 1.

As is the case with research based on an uncontrolled data source, the data is most often incomplete. This does not mean that the data identified is not of any value; it merely reflects the fact that the nature of any particular study dictates the amount and the type of data required, and that these requirements vary. It was found that the work by McDonnell Douglas Research Laboratories, (Reference 1), identified as data set 8 in Table 1, to a reasonable degree, conformed to the requirements for this study. Subsequent efforts were centered around a thorough analysis of the material in Reference 1.

Although the data base was limited, it will be seen in the following part of this report that valuable goals were derived from the available material.

TABLE 1. DATA SUMMARY

DATA SET NO.	TEST FACILITY	DATA HOLDER	DATA IDENT.	SUITABLE DATA	COMMENTS
1	OSU	OSU	NACA 0012	NO	Limited Conditions. Pressure signatures and rake drag coefficient data only.
2	OSU	USAF	Y-A W-6 BY-54	NO [b] NO [b] NO [b]	Not reviewed. The data consists of pressure signatures and rake drag coefficient data only.
3	OSU	FDL	T26622	NO [b]	Not reviewed. Data developed for 0.6 *Mach airfoil*.
4	OSU	AMES	65A413 (Basic) #1 #2 #3 #4	NO [b] NO [b] NO [b] NO [b] NO [b]	Not reviewed. Data consists of pressure signatures and rake drag coefficient data only.
5	OSU	OSU	CA(W)-2	NO	FEDD (For Early Domestic Dissemination document. Pressure signatures and rake drag coefficient data only. Drag data limited.
6	OSU	USAF	ARLTR75-0112	NO	High Reynolds' number data. Pressure signatures only.
7	DFRF	AMES	F-111	NO [b]	Three-dimensional full scale data. Rake drag coefficient data not available.

TABLE 1. DATA SUMMARY - Cont'd

DATA SET NO.	TEST FACILITY	DATA HOLDER	DATA IDENT.	SUITABLE DATA	COMMENTS
8	AMES	MDRF	NASA TM-81336	YES	Reference 1.

b - Data judged unsuitable from discussions with scientists, or otherwise found wanting.

OSU - *Ohio State University Research Laboratory*

USAF- United States Air Force

AMES- NASA Ames Research Center

DFRF- Dryden Flight Research Facility

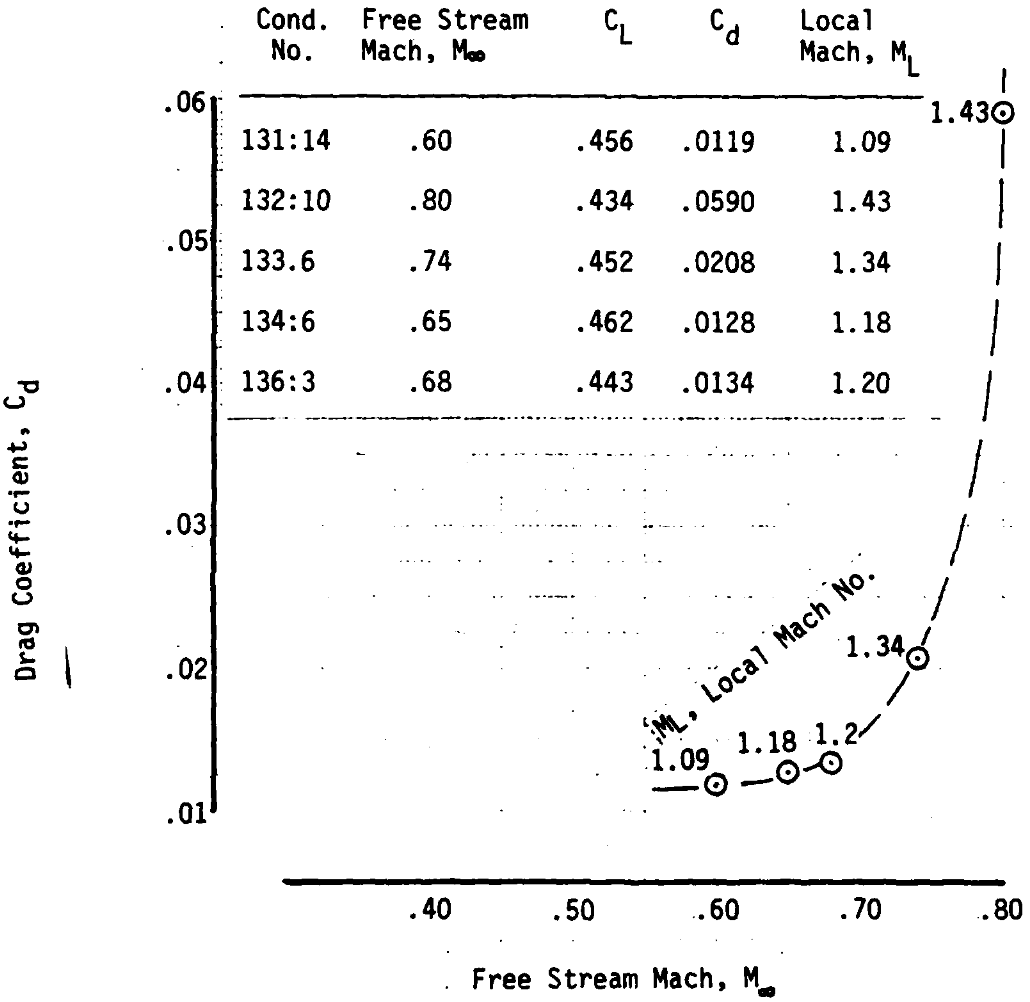
MDRF- McDonnell Douglas Research Laboratory

2. DATA ANALYSIS

The material contained in Reference 1 was thoroughly studied. The testing reported in this document was conducted on three airfoils; two supercritical airfoils, one with sharp trailing edge and one with a blunt trailing edge, and one NACA 0012 airfoil.

Although the main body of this Reference is devoted to the results from the supercritical airfoil testing, it was found that the results from the testing of the NACA 0012 airfoil exhibits some interesting characteristics. Summarized in Figure 5 are the drag values for an almost constant lift condition at varying free stream Mach conditions. Included in the figure are also the local Mach values in front of the shock for each of the test points. It is interesting to note that the drag rise occurs very close to a local Mach value of 1.3. Recalling the significance of this threshold of the local Mach value, it is reasonable to suspect an A Flow is the strong contributor in the separation at the trailing edge and thus the rapid drag rise. A more complete picture, that is the drag values for varying lift and Reynolds' numbers plotted with superimposed Mach values would indicate whether strong Mach dependent barriers exist. The boundaries of A Flow, B Flow and mixed flows could have been established by noting drag rise with Mach, Reynolds' number, or a combination of the two. Had far field wake measurements been included, the Stratford separation profile could have been evaluated for both flow conditions. Furthermore, if the boundary layer velocity profiles had been defined at several chord locations, a considerable bank of knowledge could have been assembled on the NACA 0012 airfoil. The Stratford separation profile could have been expanded or modified to include predictions of A and B Flow boundaries, and at least for the NACA 0012 airfoil, a design tool would be created.

Obviously, had this work expanded to include a large number of other airfoils this design tool should be a general guide to airfoil design to preclude premature drag rise.



Cond. No.	Free Stream Mach, M_∞	C_L	C_d	Local Mach, M_L
131:14	.60	.456	.0119	1.09
132:10	.80	.434	.0590	1.43
133.6	.74	.452	.0208	1.34
134:6	.65	.462	.0128	1.18
136:3	.68	.443	.0134	1.20

Figure 5. NACA 0012 Test Results from Reference 1.

The main body of the work in Reference 1 is dedicated to the testing and analysis of the two supercritical or Whitcomb profiles. A comprehensive test technique in support of a viscous/inviscid flow interaction study is developed in conjunction with this program. It should be noted that, in addition to pressure signatures, success was achieved in obtaining information on both boundary layer flow characteristics and the density distribution throughout the flow field. This allowed meaningful correlation of boundary layer separation with Stratfords separation criterion. This criterion plays a key role in the correlation of drag with the far wake boundary layer profile. Figure 6, which is copied from Reference 1, was included in this document for reference. If the lines representing the Stratford criterion indicate incipient separation, it can be seen that the profiles noted (a) and (b) indicate separaed flows, whereas, the (e) profile indicates an attached flow and the (c) and (d) profiles indicate flows with incipient separation. It is important to know where each of these samples occurs in the airfoil drag rise curve. Furthermore, these profiles show varying boundary layer thicknesses and the shapes themselves suggest some earlier chordwise effect to cause different profiles. It is necessary to know how each of these profiles varies with Mach, Reynolds' number, camber, thickness and surface roughness. The Stratford separation profile may be considered an early first order separation and premature drag rise prediction tool. Unfortunately the measurements from this test series did not constitute a full matrix necessary for further studies.

An extraordinary amount of work reported in the same Reference was done in the area of simulating the correct Reynolds' number conditions. The standard technique of affixing trip devices to the leading edge was explored with trip devices included on the lower leading edge surface. Drastic variations in drag characteristics were experienced with the selection of these devices as shown in Figure 7. The sensitivity to transition devices on any surface indicates global

Case	Trailing edge	M_∞	Re_c	αgeom
(a)	blunt	0.75	2×10^6	1.0
(b)	blunt	0.83	2×10^6	0.8
(c)	sharp	0.63	4×10^6	2.6
(d)	sharp	0.80	2×10^6	1.8
(e)	sharp	0.80	3×10^6	2.4

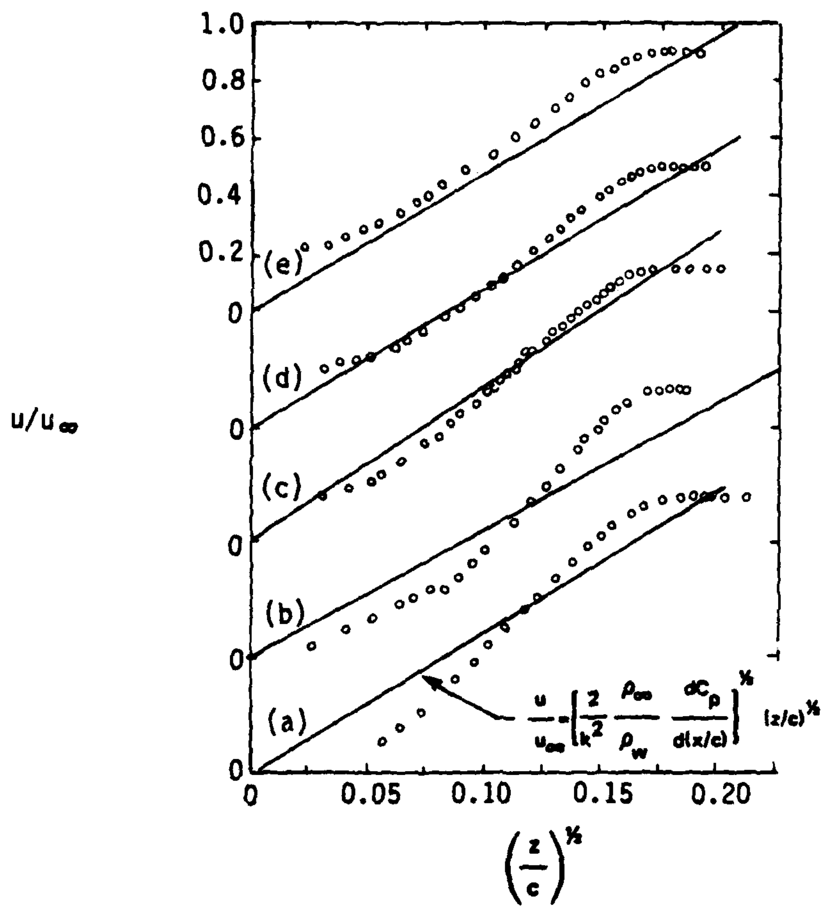


Figure 6. - Comparison of upper-surface trailing-edge profiles with Strafords separation profile. Copied from Reference 1.

Trip Configurations

x/c	k_o (mm)	$k_o/k_{o_{cr}}$ $Re_c = 3 \times 10^6$	$k_o/k_{o_{cr}}$ $Re_c = 4 \times 10^6$	
0.35	0.08	0.9	1.2	Upper surface
○ 0.06	0.05	1.0	1.3	Lower surface
□ 0.18	0.08	1.2	1.4	Lower surface
△ 0.35	0.09	1.1	1.4	Lower surface
——— Data at $Re_c = 14.5 \times 10^6$				

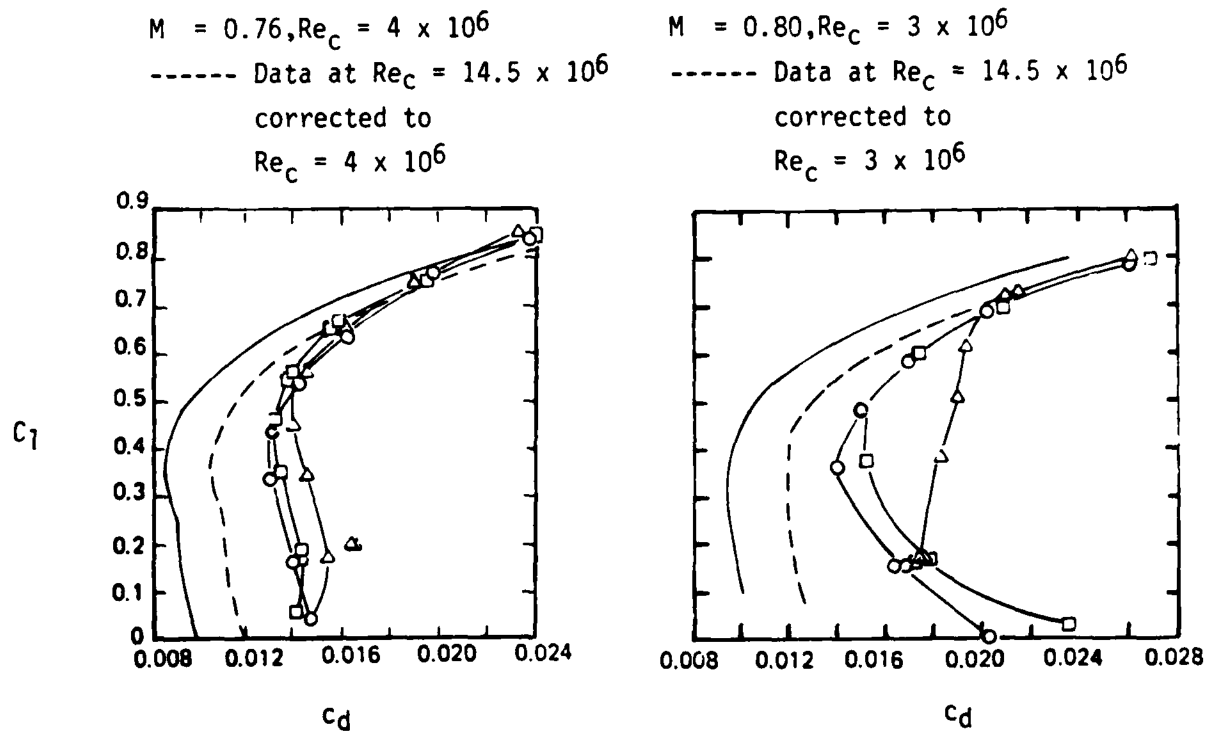


Figure 7.— Effect of lower-surface boundary-layer trip location on drag characteristics. Copied from Reference 1.

effects that render the use of such devices highly questionable in drag performance testing. Drag in the transonic regime is dependent to a large degree on boundary layer characteristics. Trip devices yield a poor simulation of full scale Reynolds' number boundary layers.

SECTION III

CONCLUSIONS

From the data and literature search accomplished by Shannon and the direct contact with researchers and scientists in the field, it was concluded that sufficient data for a correlation study of the viscous/inviscid interaction as it relates to drag creep does not exist.

Some valuable related contributions to the field of transonic testing and data collecting have been made by research facilities under contract to NASA. Private institutions probably hold proprietary data that could be useful for the subject study.

The complexity of the test procedures and the facilities required, to obtain the data that was found most valuable, suggest that the progress toward the solution to drag creep predictions will continue to be slow.

The available data allowed the structuring of a wind tunnel test program, and an analysis scheme consistent with the most advanced research in this area. Correlation between boundary layer growth, shock location and strength and the resulting drag rise is depending on such a program.

It was also concluded from the available material, that Reynolds' number is intrinsic to the drag characteristics of an airfoil. The effect of the Reynolds' number, is best not simulated via transition devices affixed to the leading edge. Only testing under full scale Reynolds' number conditions can produce valid results.

SECTION IV

RECOMMENDATIONS

It is recommended that wind tunnel testing as described below begin immediately. This study has shown that it is necessary to build the data base needed to analyze and predict the viscous/inviscid interact-ions responsible for drag creep. It has further shown that the required test capability is now available at various facilities and that senior individuals have been identified who possess the expertise to successfully complete such a program.

Discussions with one of the authors of Reference 1, Mr. F. W. Spaid confirms an interest in extending the studies in Reference 1 to higher Reynolds' number conditions. It is considered cost effective to develop a program around such a concept.

The preferred test facilities for such a program include the National Research Council (NRC) of Canada or the NASA Langley Research Center 0.3-m cryogenic tunnels. The Ohio State University Research Laboratories high pressure, high Reynolds' number tunnel is considered as an alternate test facility for a Phase II program.

Table 2 summarizes the frame work for a test program that adds to the experience in Reference 1, as well as it provides the data for a correlation analysis between drag characteristics, boundary layer growth, and upstream shock conditions.

TABLE 2 TEST PLAN

A. Airfoil Series

1. NASA 0012
2. ATA 1 (similar to DSMA 523, Sharp Trailing Edge from Reference 1)
3. ATA 2 (Advanced Technology Airfoil, profile to be defined).

B. Test Conditions

1. R_e based on chord, 6 to 40 x 10^6 (4 conditions)
2. Free stream Mach, 0.6 to 0.85 (6 conditions)
3. Lift coefficient, C_L, 0 to 0.6 (4 conditions)

C. Measurements to Include

1. Chord wise pressure signatures, upper and lower surfaces
2. Wake drag (using Far Wake Momentum Measurement Technique)
3. Boundary layer velocity profile measurements at chord stations 0.80, 0.90, 0.95 and 1.0.
4. Holographic Interferograms

NOMENCLATURE

$B^{1/2}$	-	correlation parameter
c	-	wing chord
C_L	-	lift coefficient
C_d, C_D	-	drag coefficient
ΔC_{D_C}	-	incremental drag coefficient due to compressibility
c_p	-	pressure coefficient, $(p-p_\infty)/q_\infty$
CL600	-	Canadair Challanger CL600
k	-	Vow Karman constant, 0.41
k_o	-	boundary layer trip roughness height
$k_{o_{cr}}$	-	minimum value of k_o that will cause transition to occur at the trip.
Lear 24	-	Learjet Model 24
M	-	free stream Mach number
M_L	-	local Mach
p	-	local static pressure
p	-	free stream static pressure
p_{SAM}	-	static ambient pressure
p_{SPEAK}	-	static peak pressure on airfoil
q_∞	-	free stream dynamic pressure
R_e	-	Reynolds' number
Re_c	-	Reynolds' number based on wing chord
u	-	local flow velocity
u	-	freee stream flow velocity
x	-	chordwise distance from leading edge
z	-	coordinate normal to the airfoil plane
α geom	-	angle of attack measured with respect to the tunnel test section centerline.
η	-	fraction of wing span
ρ_∞	-	density of free stream flow
ρ_W	-	density of flow at the wall

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